Analysis of the Feasibility and Utility of Radon as an Electrostatic Ion Thruster Propellant

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Abstract

The feasibility of using radon produced through decay of parent isotopes stored as fuel, as propellant in an ion thruster, is evaluated. It was believed that this approach could result in reduction of power consumption, spacecraft mass, and mission time. Analysis is done of discharge loss, power supply through Stirling convertors, propellant separation, cost, and range safety. It was found that the power and mass benefits are relatively small and heavily dependent upon the performance of technology which is not yet available. Further, absent significant improvements in isotope separation methods, the cost would be astronomical (on the order of that of the entire rest of the mission), and there would be serious safety concerns. Accordingly, it is determined that the concept is not feasible, and does not merit further study at this time.

1 Background, Motivation, and Significance

The inability of spacecraft to carry out multiple missions, and the exponential increase in propellant required as payload mass is increased, are two fundamental problems caused by the relatively low specific impulse of chemical propellants. These challenges can be overcome with electric space propulsion systems, which use various electric and magnetic phenomena to accelerate small quantities of propellant to extreme velocities, resulting in specific impulse values on the order of 10^3 seconds, albeit with thrust-to-weight ratios much less than one. Combining these propulsion systems with conventional chemical rockets to transport them to Earth orbit makes exploration and (potentially) commercial missions more effective and less costly [1].

Electrostatic ion thrusters operate by ionizing a gaseous propellant positively through energy input, then applying a strong electric field to the ions to eject them from the spacecraft at very high speed. (After they exit, an electron beam neutralizes the ions to prevent an accumulated charge imbalance from counteracting the acceleration created.) Three properties critical to the suitability of a substance for use as an electric propellant are ease of ionization, expressed as the first ionization energy per unit mass of propellant (as the second and further ionization energies are much greater, only one ionization is used); mean atomic mass, as a lower value of this results in a greater exit velocity hence greater I_{sp}; and energy required to vaporize (boiling point, and specific heat), as evaporation of the substance is necessary prior to ionization [2]. Unfortunately, elements with low ionization energies are generally metals, and light elements that vaporize readily have very high ionization energies. Additional critical considerations include chemical reactivity, ease of storage, availability/cost, and safety [2].

Mercury (Hg) was used for some initial trials because of its quasi-metallic low ionization energy and relatively low boiling point (630 K), but its toxicity and volatility caused problems [3]. Xenon (Xe), while a noble gas and therefore more energy-intensive to ionize, made up for that with a boiling point of 165 K, which eliminated the need to add energy for vaporization. However, this means that it must be stored in gaseous form, at a significant mass, volume, and reliability penalty

[4]. Hence, the search for preferable alternatives continues. Tests using krypton (Kr), which has a much lower cost than Xe, experienced lower efficiencies, as well as severe and rapid degradation of the discharge chamber and screen grid, due to the higher voltage required [8]. Analysis of the properties of Kr and argon (Ar) confirmed that using them would be less efficient [9]. Research on iodine [4] and polyatomic ions is in progress, but may or may not bear fruit. An additional fundamental problem with electric propulsion is that the propellant and fuel are not the same substance (as they are with chemical fuels); the resulting need for a separate energy source reduces the performance benefits.

Radon is the most dense naturally occurring element in period 18. Its ionization energy is less than that of all the other noble gases, and the atomic mass of the radon-220 isotope is approximately 65% more than xenon's average, which further decreases the energy required to ionize a given *mass* of propellant (to less than that of iodine). Furthermore, as a noble gas, it is chemically inert. These properties are excellent for an electrostatic ion thruster propellant; unfortunately, as its name suggests, it has the serious flaw that all of its isotopes are radioactive.

The use of radon has been dismissed as an aside in several papers in the course of discussing propellant options [2] [9], which assert but do not argue that the propellant used should not be radioactive. The fact that the most stable isotope of radon has a half-life of less than four days would seem to be a fatal objection to its use, as those authors suggest. However, as many homeowners are all too aware, radon is produced by the decay of isotopes of more massive elements.

It is hypothesized that use of Rn propellant produced through nuclear decay of parent isotope(s) stored on the spacecraft could provide propellant for an electrostatic ion thruster more efficiently and effectively than existing methods, with the energy released by the nuclear decay used to provide electrical power.

2 Methodology

To determine the benefits of radon use, the properties of radon as a electrostatic ion thruster propellant were compared to those of xenon, with the new "NASA Evolutionary Xenon Thruster" (NEXT) being used. Thrust, specific impulse, efficiency, and the various causes of discharge losses were evaluated. Next, nuclear decay series were analyzed to determine which decay processes produce radon over a useful duration. The use of a centrifuge to separate Rn gas from a very fine powder was evaluated, and the centrifuge was designed based upon storage area and mechanical and thermal loads. This was included in subsequent evaluation of the effects of this approach on the mass and power required to carry out a scientifically compelling Uranus orbiter mission under study by NASA [29][5], using energy released from the nuclear decay for electric power. The cost and range safety issues were considered. If otherwise feasible, conceptualization and evaluation of specific potential applications for this technology would have been considered, and details of a workable design developed.

NASA's proposed Uranus mission would use three NEXT thrusters, each drawing approximately 6.8 kW, with an average power use of 14 kW (operating two thrusters)[5]. At full power, NEXT operates with a beam current and voltage of 3.52 A and 1800 V respectively, and exhibits a mass utilization efficiency of about 0.78. [28] The Uranus mission is used for comparison because DAWN and Bepi-Colombo (ESA) stayed too near Sol for this approach to make sense and/or used technology that is now obsolete [6][7]. The Uranus mission would travel much further from Sol, where sunlight is diminished and thermal convertors are more favorable.

Further details of the methodology are discussed individually in the results section.

3 Results and Analysis

This section begins with discussion of simple computational results about the performance of radon propellant, and decay processes that can produce such propellant as needed. It proceeds to discuss information gathered during design of a system for separation of radon, and a possible containment vessel for the radioactive propellant in case of booster failure. It concludes with consideration of the additional costs of this approach.

3.1 Radon Performance as an Ion Thruster Propellant

The current applied to accelerate ion thruster propellant is given by Equation 1, allowing the mass flow rate to be eliminated from the thrust equation. In the equations, M denotes the mass of a single ion of propellant, I_b and V_b are the beam current and voltage respectively, and q refers to the charge of a single ion. Because a single propellant ionization is by far the most efficient approach, it can be assumed that $q = e$, where e is the charge of one electron, specified in Coulomb. A correction for undesired multiple ionization is encapsulated in an efficiency term later in the analysis [10].

$$\dot{m} = I_b M/q \tag{1}$$

The thrust exerted by an ion thruster is then given by Equation 2, with the mass flow rate encapsulated in I_b [20]. The term $0 < \gamma < 1$ is a correction factor for the combined effects of beam divergence and multiple ionization, and is generally 0.9 or more [21].

$$T = \gamma I_b \sqrt{2MV_b/e} \tag{2}$$

The energy applied to ionize the propellant sometimes causes two or more electrons to be ejected from a single atom, i.e., multiple ionization. This is an undesirable process, because it wastes power and reduces thrust and specific impulse (note the effect on the thrust equation, and by extension the I_{sp} equation, if e is replaced by $2e$). For current ion thrusters using xenon, such ions represent less than one-fifth of those that are successfully expelled [10]. It is reasonable to assume that the frequency of this phenomenon has some inverse relationship to the second and higher ionization energies of the gas involved. The first three ionization energies of the propellants under consideration are given in Table 1 (the occurrence of Xe^{+4} and above is negligible).

Table 1: Ionization Energies of Potential Propellants

Element	1st (kJ/mol)	2nd	3rd
Xenon	1170	2046	3099
Krypton	1351	2350	3565
Radon	1037	1930	2890

The second ionization energy of radon is only 6% greater than that of xenon, and the first and second combined is only 8% greater for radon than xenon. There is no rigorous theoretical basis for determining the prevalance of multiple ionization. Without any empirical data on this for radon, and given the small difference in ionization energy and the relatively small effect of multiple ionization, it is assumed that there is no significant change in multiple ionization between xenon, iodine, and radon. The correction of radon for this is included in the efficiency term of the power analysis. Krypton's high ionization energy would tend to reduce its multiple ionization, but is also why a destructively high voltage is required to ionize it [8].

With no substantial change in γ, and only M changing between xenon and radon for a given ion thruster design (hence, constant V_b and I_b), the thrust exerted by radon propellant can be compared to that from an equivalent xenon mass with Equation 3.

$$T_{\mathrm{Rn}}/T_{\mathrm{Xe}} = \sqrt{M_{\mathrm{Rn}}/M_{\mathrm{Xe}}} \tag{3}$$

Xenon has numerous stable isotopes with significant abundance in the environment. On a macro scale, it can be assumed that the mass of a given xenon ion is equal to the average, which is 131.29 AMU. As discussed below, the only isotope of radon which is produced through a suitable decay process is Rn-220, with atomic mass 220.01 AMU. Substituting these values determines that $T_{\mathrm{Rn}}/T_{\mathrm{Xe}} = 1.295$. [36]

Radon produces more thrust than xenon, though the same energy is applied to each ion, because its use increases the mass flow rate. However, the beam voltage is the limiting factor for thrust, and it is desirable to increase an ion thruster's thrust while keeping the specific impulse high, because low thrust increases both the necessary Δv for a given manuever and the mission time. Despite the higher ideal I_{sp} of krypton, krypton propellant did not in practice demonstrate significantly improved I_{sp}, because of increased losses [8].

Specific impulse in an ion thruster is calculated with Equation 4.

$$I_{sp} = \frac{T}{\dot{m}_i g} = \frac{v_i \dot{m}_i}{g \dot{m}_p} = \frac{v_i \eta_m}{g} \tag{4}$$

This is the standard equation for specific impulse, modified to account for the fact that not all of the gas which enters the anode chamber becomes ionized or is accelerated properly (further discussion of this below). The η_m term is the thruster's mass utilization efficiency, which is the proportion of the propellant used that is ionized and accelerated.

Substituting in the thrust equation, simplifying, and substituting the values of g and e in SI units gives Equation 5, which can be used to calculate the specific impulse, in seconds. This is on the order of 10^3 for typical propellants. The relative ideal specific impulses and ideal thrusts for the propellants examined are given in Table 2.

$$I_{sp} = 1.417 \cdot 10^3 \gamma \eta_m \sqrt{V_b / M_a} \tag{5}$$

Table 2: Ideal Relative Properties of Propellants, with Constant P_b

Propellant	T/T_{Xe}	$I_{sp}/I_{sp,\mathrm{Xe}}$
Xenon	1.000	1.000
Krypton	0.799	1.251
Radon	1.295	0.772

The effective specific impulse of the radon is affected by two additional considerations that are not applicable to other propellants. First, some of the radon will undergo another decay (to the short-lived polonium-216) before it can be used. As discussed in Section 3.4, this should not be significant. Secondly, in the course of decay, the mass of each atom decreases from 228.03 AMU (for both Ra-228 and Th-228) to 220.01 AMU, reducing the overall mass by a factor of 0.965 [16]. This loss reduces both thrust and specific impulse, changing $I_{sp,\mathrm{Rn}}/I_{sp,\mathrm{Xe}}$ to 0.745. The exact value of specific impulse is a function of beam power supply, discussed in the subsequent subsection.

The other differentiator of radon is discharge loss, which is the power required to ionize the neutral gas before it can be accelerated. Discharge loss has four causes: ionization (the energy directly used to ionize atoms), excitation (loss caused by electrical energy raising electrons to an excited state, but failing to eject them from their atoms), ion loss (the loss of the energy put into ions, caused by them striking the discharge chamber walls or the accelerator grids), and electron loss (similar to ion loss, but with the ionizing electrons). [20]

The ionization loss is exactly directly proportional to the propellant's first ionization energy. The ion and electron losses are characteristic of the thruster configuration and are not affected by which propellant is used. For excitation loss, data on the product of each atom's ionization cross-section with the reaction rate coefficient, for all possible excited states, is necessary.[20] This data has been determined empirically for xenon, but because of the lack of apparent utility, and the practical difficulty of chemical experimentation with radon, no such research has been done for radon. Lacking any way to carry out such experiments, an approximate method was needed.

Because excitation loss occurs when electrons jump to a higher energy level but do not escape, it can be expected to change with the first ionization energy: a lower value of that allows electrons, that would otherwise have absorbed only enough energy to enter excited states, to escape. In other words, the valence electrons of radon are already closer to escaping than those of xenon. For relatively small changes in first ionization energy, the excitation loss was modeled as also directly proportional to the first ionization energy. The accuracy of this model was checked through comparison of its predictions for discharge loss in ion thrusters operated with krypton and mercury propellants, with empirical data for those cases.

To do this, the relative proportions of the causes of discharge loss are determined from the ideal thruster equations. For a given thruster configuration, the magnitude of each form of discharge loss is a function of the mass utilitization efficiency. In general, $\dot{m} = I_b\sqrt{2MV_b/e}/(I_{sp}g_0)$ can be derived from the thrust equation, where $\dot{m}$ is the mass flow rate, M is the mass of a single propellant atom, V_b and I_b are the beam voltage and current, e is the charge of a single electron, I_{sp} is the operating specific impulse, and g_0 is the standard gravitational acceleration (9.81 m/s^2) [20].

The mass utilization efficiency of NEXT can then be determined with Equation 6 [20].

$$\eta_m = I_b M/(e\dot{m}) = \sqrt{M/(2eV_b)} I_{sp} g_0 = \sqrt{\frac{2.1801 \cdot 10^{-25}\ \text{kg}}{2 \cdot 1.602 \cdot 10^{-19}\ \text{C} \cdot 1800\ \text{V}}}(4100\ \text{s})(9.81\ \text{m/s}^2) = 0.782 \quad (6)$$

For an ion thruster which uses electron confinement to reduce the effective anode area, as NEXT does [28][22], the discharge loss magnitudes are given in Goebel and Katz [20]. With these proportions known, the accuracy of the "ideal thruster" and "proportional excitation" approximations can be evaluated. Data on the operation of NEXT indicates that its total power use exceeds beam power by 514 W while operating at 3.52 A, yielding $\eta_d = 146.023$ W/A [28]. Table 3 shows this datum for each ion thruster, under the condition of 3.5 A beam current and $\eta_m = 0.78$, with graphical interpolation used if exact data at that point was not available in the relevant paper [3][8][14][24]. It is assumed, based upon the results of comparative analysis, and that electron confinement was already being used in the 1990s [23], that it is also used on the krypton and mercury thrusters.

Table 3: Discharge Loss (η_d) Modeling

Propellant	**Modeled Discharge Loss**	**Actual Discharge Loss**	**Percentage Error**
Xenon	146.0 W/A	146 W/A	0%
Krypton	165.0 W/A	180 W/A	+9.1%
Mercury	128.8 W/A	130 W/A	+0.9%

The accuracy of this is sufficient for the first-order analysis in feasibility study. Proceeding with this assumption, the η_d change can be determined through the data of Table 3.1, with η_d specified in watts per ampere of beam current.

Though there are other power drains associated with ion thruster operation, they are negligible

Table 4: Discharge Loss in W/A

Form of Loss	**Proportion**	**Xenon Value**	**Adjustment Factor**	**Radon Estimate**
Ion loss	3.7%	5.4	1.000	5.40
Electron loss	12.0%	17.5	1.000	17.52
Ionization	24.0%	35.0	0.889	31.15
Excitation	60.3%	88.1	0.889	78.26
Total	100.0%	146.0	0.906	132.30

compared with beam power and discharge loss. Accordingly, it is assumed that all non-beam power is that due to discharge loss. The change in discharge loss is not expected to change mass utilitization efficiency, because the latter drives the former [20].

The total power consumption of the NEXT thruster, using xenon, is then given by Equation 7.

$$P_{\mathrm{Xe}} = P_b + P_{d,\mathrm{Xe}} = (1800\ \mathrm{V})(3.52\ \mathrm{A}) + 3.52\ \mathrm{A} \cdot 146.0\ \mathrm{W/A} = 6850.\ \mathrm{W} \tag{7}$$

This is essentially consistent with documentation which gives the power consumption as 6860 W for this operating condition [28]. But for radon, Equation 8 applies.

$$P_{\mathrm{Rn}} = P_b + P_{d,\mathrm{Rn}} = (1800\ \mathrm{V})(3.52\ \mathrm{A}) + 3.52\ \mathrm{A} \cdot 132.3\ \mathrm{W/A} = 6802.\ \mathrm{W} \tag{8}$$

The reduction in input power is only 48 W per thruster, or 0.8% of each thruster's total power consumption, under NEXT operating conditions.

3.2 Production of Radon and Electric Power

Radon has been known for over a century, although it was not initially understood that isotopes of radon produced through different decay processes are the same unique noble gas. Radon has atomic number 86, and is the most massive naturally occurring noble gas. Most importantly here, radon has no stable isotopes, and the least unstable isotope (Rn-222) has a half-life of only 3.8 days. This is obviously unsuitable for a spacecraft's low-thrust propellant.

Radon in the environment is produced through nuclear decay of daughter isotopes of naturally occurring uranium and thorium isotopes, which are unstable but have very long half-lives. Radon can also be produced through fission of larger atoms, or neutron capture followed by beta decay, but neutrons for either purpose cannot be produced efficiently. The best isotope for neutron production, californicium-252, generates neutrons in only 3% of its decay events [11], and plasma-based neutron generators require excessive power [12].

Because nuclear decay occurs uniformly throughout a mass, the use of a solid block of fuel material would result in the vast majority of radon atoms remaining trapped within the block until decay. Some mechanism to rapidly separate the radon from the fuel and bring it into the ionization beam is needed. This paper proposes processing the fuel into nanoparticles, i.e., an extremely fine powder with less than thirty atoms per grain, before loading it into the spacecraft. Methods for doing this are known for titanium dioxide, among other substances [13]. This fine powder would then be placed in a trough along the inside of a centrifuge. Because the gaseous radon would be much less dense than the solid particles releasing it, and the fine particles would behave like a liquid under the centripetal acceleration, a buoyant force would be exerted on the radon atoms, driving them towards the axis of the centrifuge. Once clear of the trough, they could be ionized and accelerated (see Section 3.4).

A suitable fuel isotope's atoms must have a mean time to decay to radon on the order of 5 to 100 years. If it is less than this, the change in the amount of radon and power supplied will be too large over the course of the mission, and a significant amount of propellant will be lost to decay prior to and immediately following launch. If it is more than this, most of the fuel will not decay to produce radon during an interplanetary mission, resulting in a large amount of wasted fuel and unfavorable mass tradeoffs. In addition, the fuel's decay process must produce a radon isotope with a half-life sufficient to allow almost all of the radon produced to be expended before it decays and can no longer be ionized. The fuel must not have chemical properties that make manipulating it so dangerous as to be infeasible (as is the case with elemental francium). Finally, it is desirable that the decay chain include several alpha decays, with reasonably long half-lives of subsequent elements, so that propellant particles about to produce more radon will be sorted towards the centrifuge axis, allowing the radon to escape more quickly and reducing radon decay loss.

Table 5 lists all of the isotopes of radon with half-lives greater than 20 seconds, with the half-life of each isotope, and the suitable decay processes (if any) specified [16][17]. Isotopes with very short half-lives are emitted from the decay process listing. Radon-219 is added to this table because it is produced by an otherwise good candidate decay chain.

Interplanetary missions generally have a design operational life of less than twenty years, and Earth satellites generally cannot continue to operate longer than that and/or become obsolete within that time. Accordingly, use of U-232 would result in the vast majority of the fuel mass being wasted, prohibitively increasing the necessary mass. The same problem arises with Ac-227, with at least half of the fuel being wasted; furthermore, the short half-life of the radon isotope it produces would result in serious loss of propellant to decay prior to ionization and ejection.

The decay chain from radium-228 is much more promising. The half-lives are such that most of the propellant would be used in the course of an interplanetary mission. Furthermore, the following of Ra-228 by Th-228 can be used to reduce the change in the propellant production rate in the initial years of the mission: with some of each, the former will replenish the latter as it decays.

Table 6 shows all relevant isotopes in the nuclear decay series which produces Ra-228 and Th-228, which is also known as the “thorium series” [15].

Because thorium-232 is naturally occurring, the radium and thorium fuel isotopes could possibly be extracted rather than artificially produced. In the centrifuge, particles containing one or more atoms of Ra-224 will be slightly less dense than those without, possibly moving them toward the axis and facilitating escape of the radon after the subsequent decay.

Two alpha decays occur in this series. The alpha particles can be expected to capture electrons and become helium-4 nuclei, which are much less dense than radon, so will be propelled to the axis of the centrifuge. This is undesirable because the ionization energy of helium is too high for it to be used as propellant [18], so it may interfere with operation of the thruster by absorbing beam

Table 5: All Potential Radon Isotopes

Mass Number	**Half-Life**	**Production**
203	44.2 s	No long-lived parent isotopes exist
204	70.3 s	No long-lived parent isotopes exist
205	170 s	No long-lived parent isotopes exist
206	340 s	No long-lived parent isotopes exist
207	555 s	No long-lived parent isotopes exist
208	24.4 min	No long-lived parent isotopes exist
209	28.5 min	No long-lived parent isotopes exist
210	2.4 hrs	No long-lived parent isotopes exist
211	14.6 hrs	No long-lived parent isotopes exist
212	23.9 min	No long-lived parent isotopes exist
219	**3.96 s**	Ac-227 (21.8 yrs) to Th-227 (18.7 days) to Ra-223 (11.4 days) to Rn
220	55.6 s	Ra-228 (5.75 yrs) to Th-228 (1.91 yrs) to Th-224 (3.6 days) to Rn; also U-232 (68.9 yrs) to Th-228
221	25.7 min	all parents either short-lived, or with very long-lived on chain
222	91.8 hrs	all parents either short-lived, or with very long-lived on chain
223	24.3 min	No long-lived parent isotopes exist
224	107 min	No parent isotopes exist
225	279 s	No parent isotopes exist
226	7.4 min	No parent isotopes exist
227	20.8 s	No parent isotopes exist
228	65 s	No parent isotopes exist

Table 6: Thorium Series

Isotope	**Half-Life**	**Usable Energy Per Atom**	*Comment*
Th-232	$1.41 \cdot 10^{10}$ years	0	Naturally occurring
Ra-228	5.75 years	0.046 MeV	Fuel
Ac-228	6.1 hours	2.124 MeV	Negligible decay time
Th-228	1.91 years	5.520 MeV	Fuel
Ra-224	3.6 days	5.789 MeV	May improve Rn release
Rn-220	55 seconds	0	Propellant
Various	small	0	Substance ejected
Pb-208	∞	0	Stable soon after ejection

electrons or dissipating energy through collisions. The large difference in density between radon and helium may separate them naturally within the core, with the helium accumulating at the center and the radon around it, in which case the ionization beams can merely be placed appropriately. If not, some additional mechanism for separation of the helium would be necessary.

For missions of a duration currently typical, this decay series is the best.

3.3 Heat and Electricity Generation

The heat released from the decay of the radium, thorium, and intermediate isotopes can be converted to electrical power through either a solid-state thermoelectric generator, or a Stirling heat engine. The former was used on past deep-space probes because the absence of moving parts allowed them to function reliably for decades without maintenance. In the last decade, engineers have finally succeeded in developing a working-fluid convertor resilient enough for use on long space missions: the Advanced Stirling Radioisotope Generator (ASRG). Using the heat this way can in principle reduce or eliminate the need for solar arrays and other radioisotopes.

The SEP stage of the proposed Uranus mission would use two UltraFlex, single-axis pointed, solar arrays [29]. The specific power of these arrays is 150 W/kg at 1 AU from Sol [30], diminishing with the inverse square law at greater distance. This is to be compared to the mass requirements for the heat conversion systems. The specific power of solid-state systems has been less than 3 W/kg, which cannot compete with solar energy [29].

The Advanced Stirling Convertor (ASC) is the fundamental component of the ASRG. It has demonstrated a specific power of 7 W/kg, but optimization is still in progress [25]. Furthermore, a large portion of this is the mass of the plutonium power source and its GPHS (General Purpose Heat Source) module, which includes a multilayered containment structure for the plutonium. Sunpower Incorporated, with NASA funding, has developed a new heat engine for space applications, which has demonstrated specific power 90 W/kg [26]. This technology was not ready for use by the New Horizons or Cassini missions, and there has been no further funding or demand for this type of system since then. Accordingly, at 1.29 AU and outwards, the thermal convertor has a higher specific power than the solar arrays. The spacecraft spends most of its time in this region [29], so use of the convertor is justified.

The energy release per atom, in the chain from radium-228, totals 13.479 MeV, with thorium-228 supplying only 11.309 MeV. Because most of the propellant requirement is in the first few years of operation, it is conservatively assumed for this purpose that thorium-228 makes up 90% of the propellant, for an average energy of 11.526 MeV. The total available energy over the course of fuel use is then given by Equation 9, where the first efficiency term refers to the Sunpower convertor's efficiency, and the second to the loss from non-decayed or partially decayed propellant at the end of the mission. E_{atom} is expressed in MeV per atom.

$$E_{tot} = \eta_{conv}\eta_{decay}E_{atom}\frac{\text{J}}{\text{MeV}} \cdot \frac{\text{atoms}}{\text{mol}} \cdot \frac{\text{mol}}{\text{kg}} m_{prop} =$$
$$0.32 \cdot 0.962 \cdot 11.526 \cdot 1.602 \cdot 10^{-13} \cdot 6.022 \cdot 10^{-23} \cdot 4.3854 \cdot 361.75 \text{ J} = 5.4303 \cdot 10^{14} \text{ J} \quad (9)$$

The travel time to Uranus is 4768.3 days, or $4.120 \cdot 10^8$ seconds [29]. Hence, the maximum average usable power over that time period is 1.32 MW. This will, of course, drop over the course of the mission, but at the end of that time, with only about two percent of the propellant remaining, 26.4 kW is still available. As demonstrated above, this is almost twice the requirement of the ion thrusters at full power. Power will obviously be needed by the orbiter after arrival, but at a level orders of magnitude less than that of the ion thrusters, and the small fraction of fuel remaining will continue to produce adequate power for orbiter operations. Some means of adjustable heat management will be needed, so that excess heat can be dumped to space in the early stages of the mission, but retained later in the mission to keep the convertor hot-end temperature high.

With access to this much additional power, further acceleration of the ions would be possible through use of potential differences between multiple additional grids. This would drive the specific impulse even higher, increasing Δv and allowing a faster trajectory to Uranus (which would tend to increase scientific and political support for the mission, among other benefits). Concepts such as VASIMIR and the dual-stage 4-grid make use of this principle: the latter consumes 250 kW but has a specific impulse of about 20,000 seconds. However, generally, the specific impulse benefits are overwhelmed by the necessary mass of electricity generators which, unlike the propellant, do not decrease in mass during thrusting. The same problem is apparent here: at 90 W/kg, supplying 250 kW would require 2.8 metric tons of convertors, far more than the mass of propellant. However, at lower energy requirements, it could be favorable to increase the beam power.

The proposed Uranus probe would be equipped with three NEXT units. As the distance from Sol changes, the number which can operate at a given time varies from one to all three. With a mean power supply of about 14 kW, the effect is essentially the same as that of two thrusters in continuous operation (as the propulsion system consumes the overwhelming majority of the power supplied) [29].

With the assumption that the beam voltage and current increase by the same ratio as the overall power increases, and that $P_b = \frac{1}{2}\dot{m}v_{ex}^2/\eta$, where η is found from empirical data to be 0.846, it is possible to determine the total convertor mass needed at any beam power level. This begins with the known values of discharge loss, beam power, and speciific impulse for a standard NEXT operating with Rn propellant. Iterating this procedure, in combination with others, the dry mass (i.e., final mass m_f) of the spacecraft is found. At 100% efficiency, the effective exhaust velocity would equal $\sqrt{2 \cdot P_{beam}/\dot{m}}$. Empirical data on v_{ex} with the NEXT exists, allowing the overall efficiency of the hardware to be determined. Then, for a range of beam power levels, the total power required (including discharge losses) can be calculated, and the known specific power can be used to find the generator mass. Then, the Tsilokovsky rocket equation is used to find the propellant mass needed to carry out the mission at the respective m_f and I_{sp}. Combining these results allows for optimization of power supply. The code used to do this is given in Appendix A, and the result is shown in Figure 1.

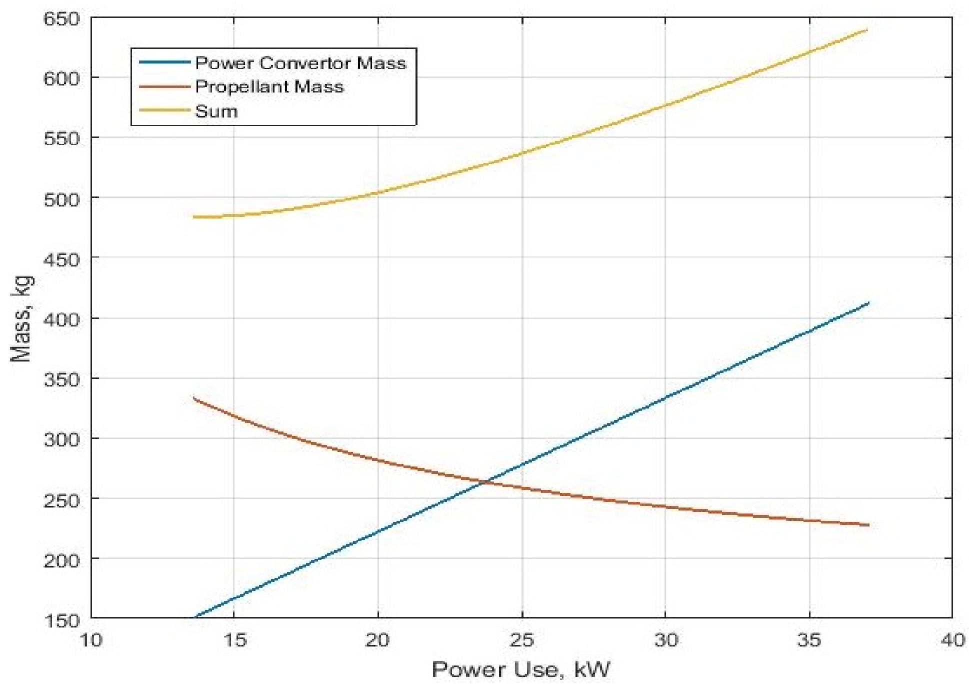

Figure 1: Power Supply Optimization

As shown, the optimal power level is almost exactly the same as what is currently used. For even small increases in power, the extra convertor mass outweighs the fuel reduction. At a lower thrust level, this would not necessarily be the case, but the nuclear decay process makes it impossible to reduce the rate of fuel use. Hence, the analysis procees with 13.6 kW being produced by 151 kg of convertors, with an effective I_{sp} of 3123 seconds resulting in a propellant mass of 332.7 kg. As with NEXT, the beam operates at 1800 volts and 3.52 amperes.

The surfeit of heat suggests its use to raise the hot-end operating temperature, driving up the

Carnot efficiency and raising the specific power. However, the design hot-end temperature of the Sunpower convertor is already 923 K, hot enough to cause significant issues with thermal loading and reduced structural strength. As the design absolute temperature ratio is 2.6, the Carnot efficiency is 61.5%. The overall efficiency of the Sunpower convertor is 50% of this, reducing the performance benefits of increased T_H by half. Elevating the already extreme temperature to the ASRG's hot-end temperature, 1123 K, presumably the upper limit at present (because of the priority of mass reduction in space applications), would only improve Carnot efficiency to 68.5%, and overall efficiency to 34.3%, improving the specific power by at most 7%. Although this is an important research area, it is beyond the scope of this paper, and it is assumed that the convertor must operate as specified by its developers.

3.4 Separator Design and Containment Vessel

To separate the radon from the fuel efficiently, a centrifuge system is needed. Figure 3.4 shows the configuration of the storage structure; the actual centrifuge shaft and rotor can be made from standard parts, with mounting through the intersections of the cross-braces. The internal annular trough within the raised rims contains the fuel, which is kept in the trough by its centripetal acceleration.

The outer radius is 0.335 meters, while the inner radius within the cavity is 0.325 meters. This was determined by the discharge chamber area in NEXT; for this application the area must be increased by a factor of three because of the helium produced by the alpha decay. In principle, with the radon more than fifty times denser than the helium, the buoyant force will rapidly concentrate the latter in the central two-thirds of the cavity's area, where it will not be ionized, and will gradually drift through the grids and out the back of the spacecraft. As the alpha decay will release it at a high initial temperature, the ensuing cooling and decompression may actually add a small amount of thrust, but it could also interfere with the radon ionization or acceleration, increasing discharge and mass loss through excitation in the discharge chamber or colliding with radon as it

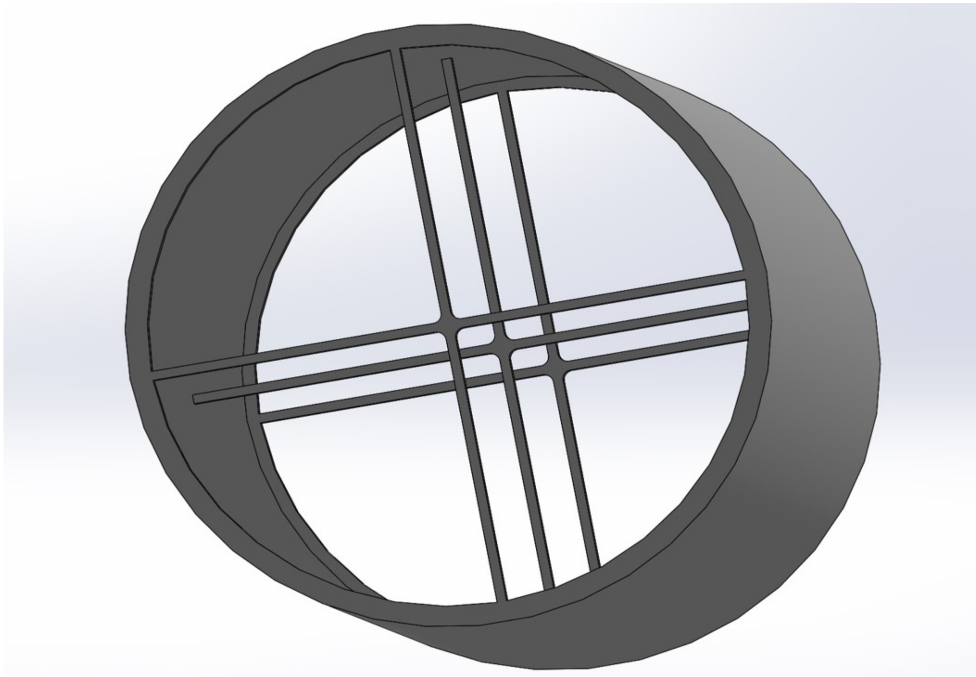

Figure 2: Fuel Storage/Propellant Separation System

passes through the grids. Given the complexity of this problem, it is left for future research.

As the solid fuel is much more dense than the gaseous propellant, the trough's radial size is small compared to that of the discharge area (the radius to the rims around the ends of the interior is 0.307 m). The unit was designed based on the assumption of an initial even split between Ra and Th; as the proportion of the denser thorium has since been increased, this adds an additional margin of safety in preventing escape of fuel, with a mean density of about 10,000 kg/m^3. The acceleration of the centrifuge should be sufficiently greater than that of the spacecraft to prevent shifting of fuel towards the back side of the trough, but some lightweight mechanism to smooth it again could be added, if this proves to be a problem.

The second program in Appendix A is used to determine the necessary angular velocity of the centrifuge as a function of the loss from decay prior to ionization and exit. It is assumed, because of the low relative velocity, and because the standard drag equation does not work properly at that length scale, that the aerodynamic drag on the radon is negligible. The density of radon gas at standard conditions is orders of magnitude less than that of the fuel [16], so it is also neglected in the buoyancy calculation. Because radon does not bond with metals, and metals have excess valence electrons, the chemical separation of radon from fuel grains should occur in negligible time. Because the rotation rates involved are so low, the loss from premature decay can be reduced to a negligible level. As shown in Figure 3, for loss of 0.1% (less than the margin of error), the angular velocity need only be 2.804 rad/s, or 26.7 RPM. This results in a total force on the outer ring of less than 100 N. With the centrifuge made of a titanium alloy (10V-2Fe-3Al solution-treated bar stock), for the combination of strength and low density that it affords, this causes strain in the range of 10^{-9}, per a SolidWorks stress analysis.

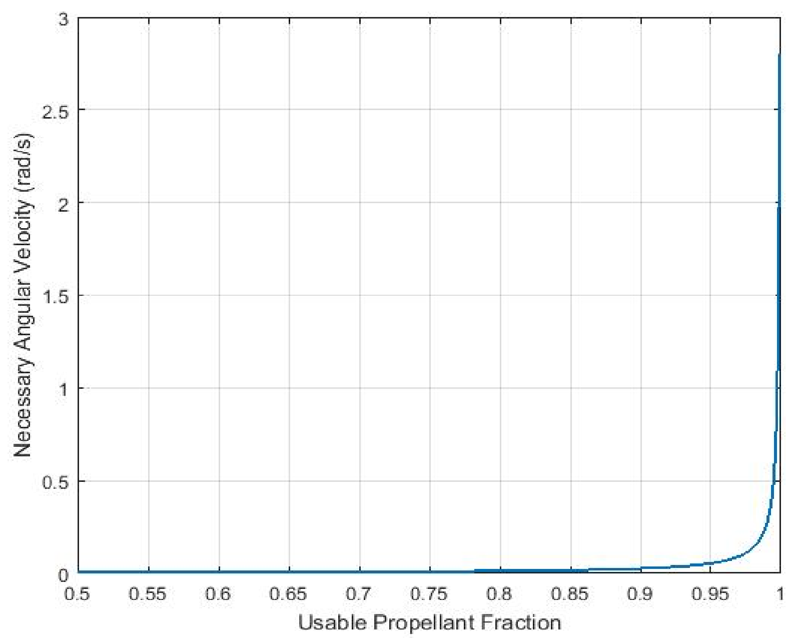

Figure 3: Radon Escape Behavior

However, the high operating temperature, while well below the melting point of titanium, still causes significant thermal stress. That is the cause of almost all of the stress shown in Figure 4. Although the scale exceeds the yielding stress, that only happens at the hubs of the supports and is an artifact of the constraints the model places them under, which should not be expected in reality.

Elsewhere, the stress is within the safe limit, although considerable improvement of this design to reduce mass and improve strength is obviously possible.

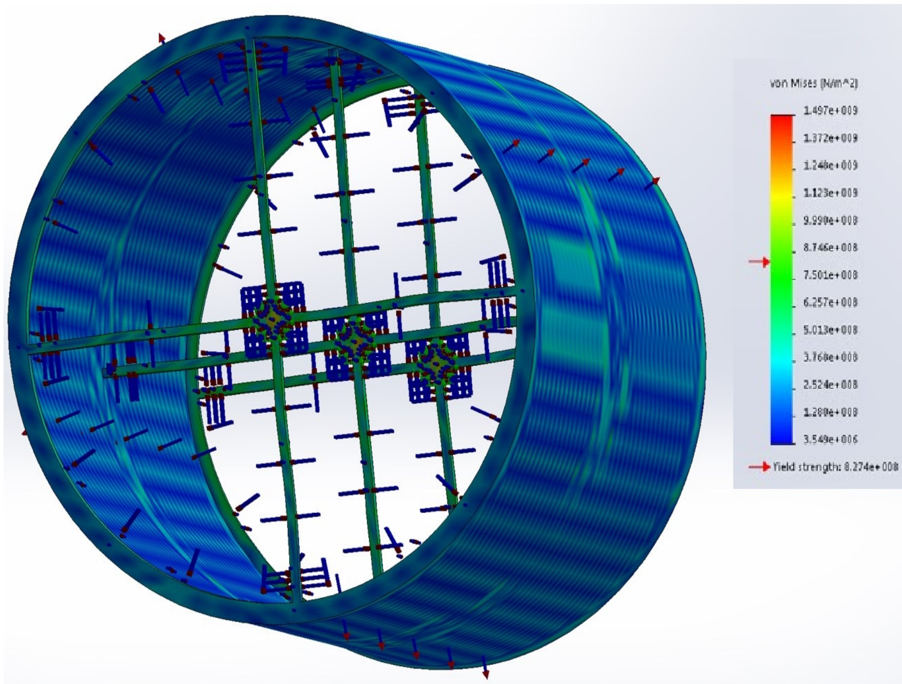

Figure 4: Centrifuge Thermal and Loading Stress

It should be noted that the detailed design of the centrifuge was determined through slightly different assumptions about the fuel density and quantity, but the effect of this should not be significant, and would tend, if anything, to reduce the necessary mass.

The low speed of the centrifuge should reduce frictional losses and minimize the power needed to maintain rotation; there should be no problems with the angular momentum causing unwanted attitude stability. The ionization beams can pass either through the cylinder along its height; or from annular cathodes, to anodes within the outer part of the cavity.

The fuel storage structure most likely must, itself, be placed within a containment vessel that would prevent an uncontrolled release in the event of booster failure, which creates extreme temperatures and forces that can only be survived with a bulky structure. Further, the approach to this used with plutonium is to seal it inside multiple layers of different substances, forming a so-called

"General Purpose Heat Source module" (GPHS module)[32]. That cannot be done here because the radon has to be able to escape, so a single large containment vessel must be used, and jettisoned after booster burnout. Each GPHS has mass 1.5 kg and generates 250 W [32], and plutonium-238 generates 0.568 W/g initially [27], implying that each GPHS consists of 0.44 kg of Pu-238 and 1.06 kg of other substances, or 2.41 kg of container for every 1 kg of radioactive material. Significant development work would be required for such an external container.

3.5 Mass and Orbit Effects

The NASA report on this mission employs a complex trajectory in which the spacecraft orbits Sol twice, operating the ion thrusters 90% of the time to gradually accelerate, and then performs a flyby of Earth that puts it on the Uranus transfer orbit, during which it cruises. The transfer trajectory is so much faster than Hohmann that deceleration relative to Sol is needed upon arrival. The trajectory is based upon the exact performance parameters of the xenon thruster, which differ from those of this concept in terms of thrust, specific impulse, and the controllability of fuel use [29]. The complexity of the problem means that simplifying assumptions must be used.

In principle, it is possible to reduce the Δv needed as the thrust increases, as seen in the difference between impulsive-burn and low-thrust orbits. However, this mostly depends on the ability to apply thrust at the points in the orbit at which doing so is most effective. Unlike with a xenon thruster, the thorium and radium cannot be saved for later: they decay constantly and inexorably. Though a higher thrust may reduce mission time, it cannot in this case reduce impulse required. Accordingly, it is assumed that the Δv budget is the same as in the NASA report: 7.0 km/s [29]. This assumption requires that a trajectory be found that makes full use of the portion of the fuel that decays during it. The fuel which does *not* do this is considered in the mass calculation below.

The total xenon propellant mass is not specified in the NASA report, but can be derived from the known NEXT I_{sp} of 4190 s, and the Tsilokovsky equation, as in Equation 10, where m_f is the remaining (final) mass when thrust concludes.

$$m_{\mathrm{Xe}} = m_f \left(e^{\frac{\Delta v}{I_{sp} g_0}} - 1 \right) = 1487.8 \text{ kg} \cdot \left(e^{7000/(4190 \cdot 9.81)} - 1 \right) = 276.2 \text{ kg} \tag{10}$$

At the end of the cruising phase, another impulse for Uranus orbital insertion is necessary. The NASA report indicates that the Δv of this manuever is 1661 m/s, and specifies that hydrazine bipropellant is used for this manuever, with an I_{sp} of 332 seconds [29]. At this stage in a radon-based mission, there would still be sufficient propellant flow to execute this manuever with the ion thruster, if the fuel distribution is proper. It should be noted that the transition from an impulsive to low-thrust deceleration is likely to increase the Δv, which would be important to fully analyze in future work.

This, and other effects that change the mass, are as follows:

- Plutonium, and its separate containment vessel, is not needed for the orbiter's RTG. The orbiter requires at least 367 W throughout its operation [29], and the convertor is only 38% efficient [25], necessitating plutonium that produces 1153 W of heat at t_0. At 0.568 W/g, including container mass at 2.41/1, this eliminates 6.9 kg.

- At the operating temperature of the spacecraft, xenon is a gas. It must be stored in a pressurized tank, which has a tankage fraction of approximately 0.1 [24][33]. The tank mass is then $0.1 \cdot 276.2$ kg $= 27.6$ kg, which can be eliminated.

- The solar arrays used to power the ion thrusters are eliminated. The total mass of the electrical power system in the SEP stage is specified as 188.0 kg, producing an average of 14 kW while the ion thruster operates. This is consistent with the UltraFlex array's documentation, given the average distance from Sol during this mission phase [30]. Accordingly, 188.0 kg is eliminated.

- For a steady supply of 13.6 kW during flight, at 90 W/kg, the Sunpower convertors add 151.1 kg to the necessary mass.

- As determined by SolidWorks, the total mass of the centrifuge part is 162.6 kg. Significant reduction of this is probably possible.

- Insertion of the payload into orbit around Uranus requires an additional Δv of 1661 m/s. From Tsilokovsky, this requires 421.7 kg of propellant, given an orbiter mass of 633.9 kg and $I_{sp} = 332$ s[29]. The engine apparatus needed for this has an estimated additional mass of 26.1 kg (including a 667-N engine bell and the propellant tank)[34][35], for a total of 447.8 kg, all of which can be eliminated.

- With arrival at Uranus 4768 days after launch, 6.8 thorium-228 half-lives have elapsed, so only 0.87% of the initial fuel should remain, or 2.9 kg. However, this would result in insufficient power and propellant for orbital insertion ad satellite operations. At $I_{sp} = 3123$ s, 35.3 kg of propellant is needed for the insertion manuever. After 10 years—1.3 combined half-lives–40.4% of the radium-228 fuel should remain, so the fuel must include 87.2 kg of radium-228 initially. 28.2% of this mass remains after the manuever is completed at Uranus, so 24.6 kg of radium or decay products thereof is unused and wasted (except as a source of power for the orbiter). The decay timing issue would need to be resolved for this to be practical.

- The mass of the centrifuge apparatus as designed is 162.6 kg.

- The sum of the above mass changes is -329.1 kilograms. The m_f for the Venus transfer manuever is then given by the sum of the component masses (entry probe, orbiter, and propulsion stage, respectively) plus the correction: $m_f = 88.87\text{ kg} + 633.89\text{ kg} + 765.08\text{ kg} - 329.1\text{ kg} = 1158.8\text{ kg}$ [29]. This value, $\Delta v = 7000$ m/s, and $I_{sp} = 3123$ s, placed in the Tsilokovsky equation, give a fuel mass of 297.47 kilograms. As discussed above, the necessary mass of xenon propellant is found to be 276.2 kg; 21.3 kilograms of fuel are added.

- The mass of the containment vessel is approximated as 2.41 times that of the radioactive material. This equates to 716.9 kilograms. This is not included in the above calculations because it can be jettisoned once the spacecraft is clear of Earth, before beginning to operate the electric propulsion system. However, it is still necessary for the booster to convey it to orbit, in which area it wipes out the mass benefits of the radon approach and then some. Reduction of this mass would be a top priority in making this a viable concept.

The above mass changes are summarized in Table 3.5. The fuel mass is calculated for the m_f that includes the centrifuge. Overall, without the containment vessel (or with its mass greatly

Change	**Magnitude (kg)**
Plutonium removed	-6.9
Xenon tank removed	-27.6
Solar arrays removed	-188.0
Thermal convertors added	$+151.1$
Uranus inserter removed	-447.8
Wasted thorium	$+2.9$
Wasted radium	$+24.6$
Centrifuge	$+162.6$
Changed fuel mass	$+21.3$
Containment vessel	$+716.9$
Subtotal without centrifuge	-470.4
Subtotal with centrifuge	-307.8
Total with cont. vessel (booster payload)	$+409.1$

Table 7: Mass Effect

reduced), and with the centrifuge mass reduced to something more reasonable, the spacecraft mass is reduced by almost half a metric ton. With a boosting cost on the order of $2500 per kilogram, this represents a potential savings of over a million dollars in this area. However, if the containment vessel is necessary, and approximately that massive, then the whole concept is inferior to existing approaches.

3.6 Safety

The fuel isotopes undergo alpha decay, making them a severe hazard if inhaled [37]. Unlike the plutonium used for past RTG units, this fuel must already be in the form of fine particles. For plutonium, an impact releasing sufficient energy to atomize the material would have been necessary to create a respiratory hazard. But for this design, any breach of the containment vessel could release dangerous radioactive material. Without the containment vessel, this could occur as the result of a wide range of booster failures events. A far larger quantity of radioactive material is necessary for this than with typical RTGs, and the short half-life means that the radiation exposure would be severe.

However, the short half-life also eliminates the threat much more quickly than with plutonium. Pu-232 has a half-life of 87.7 years [27], meaning that any area contaminated with fine particles of it would continue to be dangerous for centuries, and the particles would have all of that time to spread through wind and water [37]. The isotopes used here would be inert within a few decades; the contaminated area could simply be evacuated and re-occupied once the alpha emissions had dropped to safe levels. The serious danger is that a hard impact, or wind, could disperse the particles over an area too large to evacuate, significantly increasing the background radiation level in that region for decades, and thereby the subsequent incidence of lung cancer and other health problems.

3.7 Cost

Elemental radium no longer has any industrial applications, and is used only for scientific purposes on an as-needed basis, so there is no way to estimate costs, other than to assert that they would be very high. The direct separation of radium-228 from thorium-232 is not feasible because the half-life of Th-232 makes its concentration therein only about 0.4 ppb [17], prohibitively requiring the processing of more than 100 million metric tons of Th-232 [19]. In the 1950s, a technique was developed for separating radium-226 from barium and thorium-230 compounds [16], but radium-

228 is only a daughter of thorium-232 and francium-228 (which does not occur naturally) [17] and cannot be obtained this way. Estimates for a per-gram cost during the Second World War are around $10,000 [36], and even at a tenth of this, it would cost about one hundred million dollars to stock the spacecraft as needed, and a dedicated production facility would be needed. That is a significant fraction of the entire mission's cost [29], and would eliminate any financial benefit of this approach and then some.

Thorium-228 is a decay product of uranium-232, considered as a fuel in its own right above; as a form of nuclear waste produced by thorium reactors (among other processes), it might in principle be possible to obtain it inexpensively. It could then be processed for thorium-228, which would make up a few percent of the mass at an early time in the decay. However, the purity of the fuel would need to be extremely high. Not only is any impurity a useless dead weight that replaces necessary fuel, but impurities in xenon propellant at the 10 ppm level can cause oxidation of the electron emitters severe enough to disable the thruster [20]. The helium decay product should not be a concern in this regard, as it is the most chemically inert element that exists; however, any water or oxygen released by chemical reactions of impurities in the hot fuel could have this effect. Obtaining this level of purity is similar to the process of making weapons-grade uranium [38][39], and is likely to have a similarly extreme cost, unless some chemical means of separating thorium from uranium exists or could be developed. Low-cost thorium-228 would be necessary but not sufficient: the use of the ion thruster for Uranus orbit insertion is critical to the utility of the entire approach, and that requires about a hundred kilograms of radium-228.

An additional practical issue arises with the fuel isotopes, particularly thorium: they must be separated or produced and then very rapidly transferred to the booster and launched, to minimize loss of fuel while sitting on the pad. Given the frequency with which bad weather, mechanical problems, and other issues cause launches to be delayed, this problem could be serious, especially given the high cost of manufacturing it in the first place. In the event of a launch delay, short of running everything through the separation process again, there would be no way to selectively remove the decay product, meaning that even a delay of a week could require replacement of all of the fuel, to keep the total available impulse within acceptable parameters.

Overall, at the present state of technology, the cost appears to be an irrefutable objection to this concept's viability.

3.8 Public/Political Perception

In light of the issues presented above, even if the mass could in principle be reduced to the point where this approach reduces cost and/or mission time, it would be hard to justify funding further research into this approach. Scenarios where the benefits justify the cost are implausible.

In terms of nonclemature, this concept is a public relations nightmare. Of the four elements with significant involvement, three are known to the public only as menaces: radium as the bane of Marie Curie, radon as a sinister poison lurking in one's basement, and lead as a ubitiqous toxin that damages the brains of children. Though the actual resultant concentration is negligible, it may be difficult to explain to the public that a proposed taxpayer-funded spacecraft uses a notorious poison as propellant and spews lead across the solar system.

The level of public paranoia about the much smaller and less dispersable quantities of plutonium on previous missions of this type suggests that, with orders of magnitude more radioactive material, the pushback may be overwhelming [40]. Politicians are likely to insist on the use of the containment vessel as a condition for funding, which makes this approach inferior to existing ones, barring the development of a way to greatly reduce its mass.

An advanced authoritarian country, such as China, might have less difficulty with this. However, even China would not be immune to diplomatic pressure to discourage them from launching something which might crash somewhere else and release dangerous radioactive material. In any case, supporting such an endeavour would be ethically dubious at best.

4 Conclusions and Future Work

- The reduction in input power due to reduced discharge loss is negligible, although a lower ionization voltage might reduce susceptibility to arcing.
- The benefit in terms of mass required by the booster is dependent upon the mass of the containment vessel. Although mission time could in principle be reduced this way even with a massive containment vessel, that could be more safely and efficiently accomplished with a conventional SEP stage with extra xenon and larger solar arrays.
- With the development of a relatively low-energy chemical process capable of separating Th-228 from U-232, that could be conducted near the launch site, or some other way of inexpensively separating it; a similarly low-cost way of extracting or producing Ra-228; *and* a way to eliminate the containment vessel safely or else reduce its mass by at least a factor of five, this approach might become technically feasible.
- There are significant design challenges and uncertainties, such as the effectiveness of the separation apparatus, the orbital insertion thrust application, and the workability of some necessary equipment (such as the Sunpower convertor).
- In light of this long list of issues, even if the critical ones could be resolved, the development costs would likely exceed any financial benefit of the new approach. With nuclear thermal engines seemingly finally becoming viable, and controlled fusion within reach, the ion thruster may be obsolete soon in any event, so a large development investment in this is unwise. Also, solar arrays' specific power may improve more quickly than the heat engines'.
- Overall, this concept is not feasible, and, barring a number of highly improbable changes in various parameters, it does not merit any further investigation.
- The use of uranium-232 as a radon-producing fuel to maintain satellite orbits over a duration of centuries might make sense, as solar panels would severely degrade over such a time.

A Code

Power Optimization

```
dischargeLoss=132.3;%W/A
Isp_NEXT=3123.2;%seconds; adjusted for Rn propellant
g0=9.81;%m/s^2
specificPower=90;%W/kg
T_NEXT=0.296;%N, adjusted for Rn propellant
m_f=1215.44;%this is from the last iteration, including all components
(except rad. vessel, obviously)
%assume that increase in power is from increased I_b and V_b half and half,
%based on Ohm's law
V_b=1800:10:3000;%limits were initially at those of dual-stage 4-grid;
%domain has been reduced for better focus on the desired value
I_b=3.52:0.02:5.92;
m_dot=T_NEXT/(Isp_NEXT*g0);
P_b=V_b.*I_b;
%Correct for difference between ideal case
%(100% efficient electrical-kinetic conversion), and reality
ideal_v_ex=sqrt(2*P_b/m_dot);
actual_baseline_v_ex=Isp_NEXT*9.81;
efficiency=actual_baseline_v_ex/ideal_v_ex(1);
v_ex=ideal_v_ex*efficiency;
%Calculate total mass
totalPower=P_b+dischargeLoss*I_b;%W
powerMass=totalPower/specificPower;
baselinePowerMass=P_b(1)/specificPower;
```

```
propelMass=(m_f+powerMass-baselinePowerMass).*(exp(7000./v_ex)-1);
bothMass=powerMass+propelMass;
totalPowerkW=totalPower/1000;
plot(totalPowerkW,powerMass,totalPowerkW,propelMass,
totalPowerkW,bothMass,'LineWidth',1.5)
legend('Power Convertor Mass','Propellant Mass','Sum')
xlabel('Power Use, kW')
ylabel('Mass, kg')
grid on
```

Centrifuge Design

```
usableFuelRatio=0.5:0.001:1; %AKA N/N_0
Rn220HL=55;%Radon-220 half-life
escapeTime=Rn220HL*log(usableFuelRatio)/log(0.5);%take log base 1/2
MaxCylinderHeight=1.6;%meters; this is because of booster size limitations
fuelMass=251.5;
fuelDensity=10000;%kg/m^3
r_inner=0.3118;%m
r_outer=sqrt((r_inner^2)+fuelMass/(fuelDensity*MaxCylinderHeight*pi));
neededAccel=2*(r_outer-r_inner)./(escapeTime.^2);
omega=sqrt(neededAccel/r_outer);%angular velocity
```

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